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14. ABSTRACT

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Orbital Analysis of Macron Propulsion

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Recent advances in energy storage and solid-state switching enabled the use of peristaltic, pulsed inductive acceleration of non-ferritic particles for spacecraft propulsion. Macron Launched Propulsion (MLP) systems electromagnetically accelerate gram-sized aluminum particles (i.e. macrons) to achieve exit velocities between 5 and 10km/s, achieving specific impulses between 600 and 1,000s. Research was conducted to analyze this system's potential effects on the orbital debris environment as well as to formulate possible implementations of this technology. Ultimately, the direction, velocity and altitude at which these macrons are fired determine the macron's trajectory and dictate the level of impact upon the orbital debris environment. Research supports the implementation of the technology as a multi-purpose orbital maneuvering system but cautions the use of this system in a manner that could result macrons entering into a stable Earth orbital trajectory.

Nomenclature

ρ	=	atmospheric density	F	=	force of thrust
v	=	macron exit velocity	ΔV	=	change in velocity
C_d	=	coefficient of drag	ΔV_s	=	change in velocity for simple plane change
A	=	macron cross-sectional area	V_i	=	spacecraft initial velocity
F_d	=	force of drag	θ	=	inclination change angle
I_{sp}	=	specific impulse	m_i	=	initial spacecraft mass
g_0	=	earth's gravitational constant	m_f	=	final spacecraft mass
R	=	radius			

I. Introduction

A macron Launched Propulsion (MLP) system utilizing peristaltic, pulsed inductive acceleration techniques of non-ferritic particles to electromagnetically accelerate macrons (i.e. a macroscopic particle with an approximate mass of 1 gram[1]) to speeds between 5 and 10 km/s has recently been developed. Pulsed induction is an acceleration mechanism based on the repulsive force exerted on a magnetically induced conductor by employing a series of one or more pulse coils. Since this process is inductive, no direct mechanical or electrical connection with the projectile is required resulting in an increased overall system lifetime with high repetition rates. The maximum exit velocity of a macron fired from this system is thus determined by the intrinsic behavior of the material conductivity, its density, the induced current sheet thickness and the number of stages of the propulsion system. The maximum exit velocity achievable by this approach is independent of the magnetic waveform and is primarily limited by the material characteristics of the macron. Ultimately, heating of the macron due to the magnitude of the induced currents can cause vaporization if the material properties are unfavorable. Aluminum was chosen as the material of choice due to its excellent operating parameters and low cost. In its simplest manifestation, the physics

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of the MLP system are similar to those employed in the plasma-based Pulsed Inductive Thruster (PIT) and the Magnetically Accelerated Plasmoid (MAP) thruster.[2]

The mission benefits of a macron-acceleration based propulsion system are far reaching. Modeling suggests a highly efficient thruster with no ionization losses may achieve efficiencies upwards of 90% while operating at 600 to 1,000s of specific impulse (Isp). The primary efficiency loss in the MLP thruster is a result of Ohmic heating of the macron itself. Therefore, macrons are ejected from the thruster without significant heat transfer to the thruster's walls minimizing the amount of thermal management required. Similarly, this technology is uniquely suited to providing primary, in-space propulsion and is applicable for >10 kW level propulsion systems. These performance parameters allow for very rapid response of necessary orbital maneuvers. This system would enable multi-day orbital maneuvering as compared to the traditional multi-month electric propulsion maneuvers. The traditional benefits of a pulsed system also apply: exceptional variability in thrust and power levels, variable specific impulse and high specific power. This also implies high thrust-to-power ratios of greater than 200 mN/kW at 800s of specific impulse.[3] With these performance parameters, this technology could successfully bridge the performance gap between bi-propellant thrusters and conventional electric propulsion. However, this concept brings with it the need for comprehensive modeling of its effects on the space environment from a perspective of orbital debris. The overall goal of this research is to determine the effects these macrons will have on the debris environment, potential hazards to other orbiting spacecraft as well as, formulating possible implementations of this technology.

II. Orbital Analysis

Stray or unaccounted for macrons orbiting the Earth, with high relative velocities with orbiting spacecraft, can have a tremendous impact on space operations. In order to understand and attempt to minimize the adverse affects of orbiting macrons, simulations were performed to model their trajectories once they have been fired from a MLP system. The three primary parameters which determine the final trajectory of a macron are the direction, velocity and altitude at which they are fired. Therefore, comprehensive models were created to simulate all firing angle (i.e. 0 to 360 degrees) and velocity (i.e. 5 to 10 km/s) combinations for carious altitudes.

A. Micrometeoroid and Orbital Debris Environment

The flux of orbital debris in LEO is greater than the micrometeoroid flux. Therefore, man-made debris accounts for the majority of the LEO collision hazards. Fig. 1 shows the approximate micrometeoroid and orbital debris (MMOD) flux according to particle size in LEO. The number density of micrometeoroids in GEO is slightly less than in LEO; however, Fig. 1 is a reasonably accurate approximation regardless of altitude. The micrometeoroid flux, with units of particles/(m^2year), to a spacecraft is given by[4]

$$Flux = 3.156 \times 10^{7}[A^{-4.38} + B + C] \tag{1}$$

where m is the micrometeoroid mass in grams and

$$A = 15 + 2.2 \times 10^{3} m^{0.306}$$
$$B = 1.3 \times 10^{-9}(m + 10^{11}m^{2} + 10^{27}m^{4})^{-.0306}$$
$$C = 1.3 \times 10^{-16}(m + 10^{6}m^{2})^{-0.85}$$

The representative mass density distribution is

$$\rho = 2\frac{g}{cm^{3}},\ m < 10^{-6}g$$

$$\rho = 1\frac{g}{cm^{3}},\ 10^{-6} \leq m < 0.01g$$

(2)

$$\rho = 0.5\frac{g}{cm^{3}},\ m > 0.01g$$

Any increase in man-made orbital debris is obviously undesirable and an increase beyond the micrometeoroid flux should be avoided. In GEO, collision probabilities are much lower due to a decrease in man-made orbital

debris. Fired macrons that remain in orbit (i.e. do not reenter or escape) will adversely contribute to the debris environment.

The shaded area in Fig. 1 represents the current macron radius range under consideration in this study. Through this size range, man-made orbital debris is substantial. Above this size range, man-made orbital debris tends to dominate the particle flux. Obviously, the goal in using any macron propulsion system would be to not increase the MMOD flux at all. Thus, maneuvering scenarios that place macrons on reentry or escape trajectories are desired. Less desirable scenarios would result in macrons being placed into unstable Earth orbits which degrade quickly due to drag. Undesirable scenarios place macrons into stable Earth orbits. Subsequent sections will distinguish between stable and unstable Earth orbital trajectories.

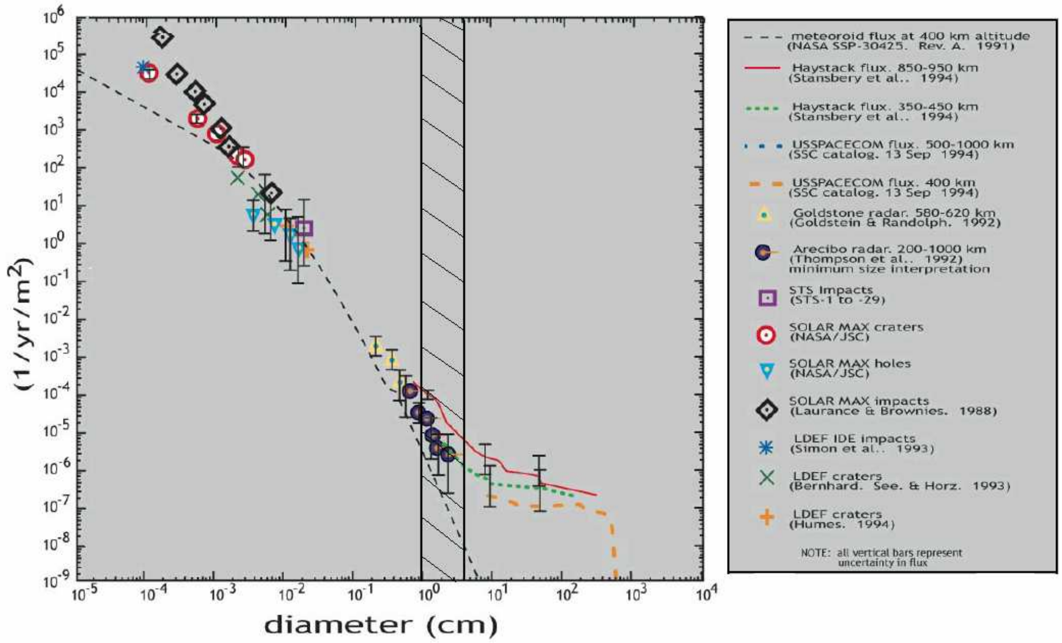

Figure 1. Approximate debris flux compared to the micrometeoroid flux in LEO by object size.[5]

B. Trajectory Analysis

Satellite Tool Kit™ (STK) was used to determine the resulting trajectories of the fired macrons. STK software is a general-purpose modeling and analysis software for space systems which incorporates spatial mechanics and integrated visualization.[6]It solves the second-order equations of motions through a variation of Cowell's formulation and Runge-Kutta numerical techniques.[7] STK simulations were designed to predict the orbit of a 1g macron being fired from a circularly orbiting satellite with an initial inclination of 0deg from various altitudes. Macrons were modeled as spheres and three primary firing altitudes were investigated: 300, 20,000 and 35,780km (LEO, MEO, and GEO, respectively). Individual simulations were created for all firing angles across a two-dimensional plane parallel to the Local Vertical-Local Horizon (LVLH) of Earth with the principle coordinate system based off of the velocity vector in the x-direction (RAM), as seen in Fig. 2. Only firing angles within this plane were simulated since maneuvering outside of this plane is typically inefficient.

Preliminary results from the orbital trajectory simulations, showed that third-body effects were generally negligible. However, at very specific firing angle, exit velocity and altitude combinations, macrons have the potential to enter into highly elliptical orbits with a radius of apogee approximately equal to or greater than the radius of Lagrangian Point 1 (R_{L1}≈3.235e5km). These orbits are referred to as Lunar Sphere of Influence (LSOI) orbits and require three-body orbital analysis. Over time, simulations have shown the majority of the LSOI orbits ultimately enter into an escape or reentry trajectory due to lunar perturbations. Furthermore, LSOI orbital effects are most notable at lower macron exit velocities (i.e. 5 to 9km/s) and are minimal at higher macron exit velocities (i.e. 10+ km/s). All trajectory analysis simulations were conducted using a standardized Epoch time. The resulting trajectories were grouped into four categories (escape, reenter, orbit or LSOI) as shown in Fig. 3. For the purposes of this study, only combinations of firing angles, exit velocities and altitude combinations that resulted in stable orbits were considered potentially hazardous.

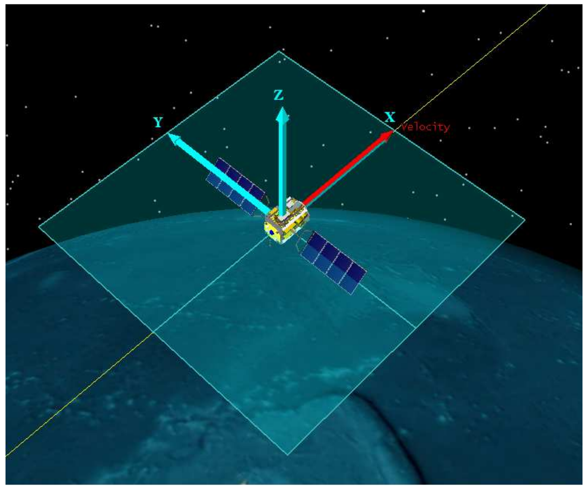

Figure 2. Macron firing plane. *The xy-plane is tangent to the Earth's surface. The z-direction points in the radial direction.*

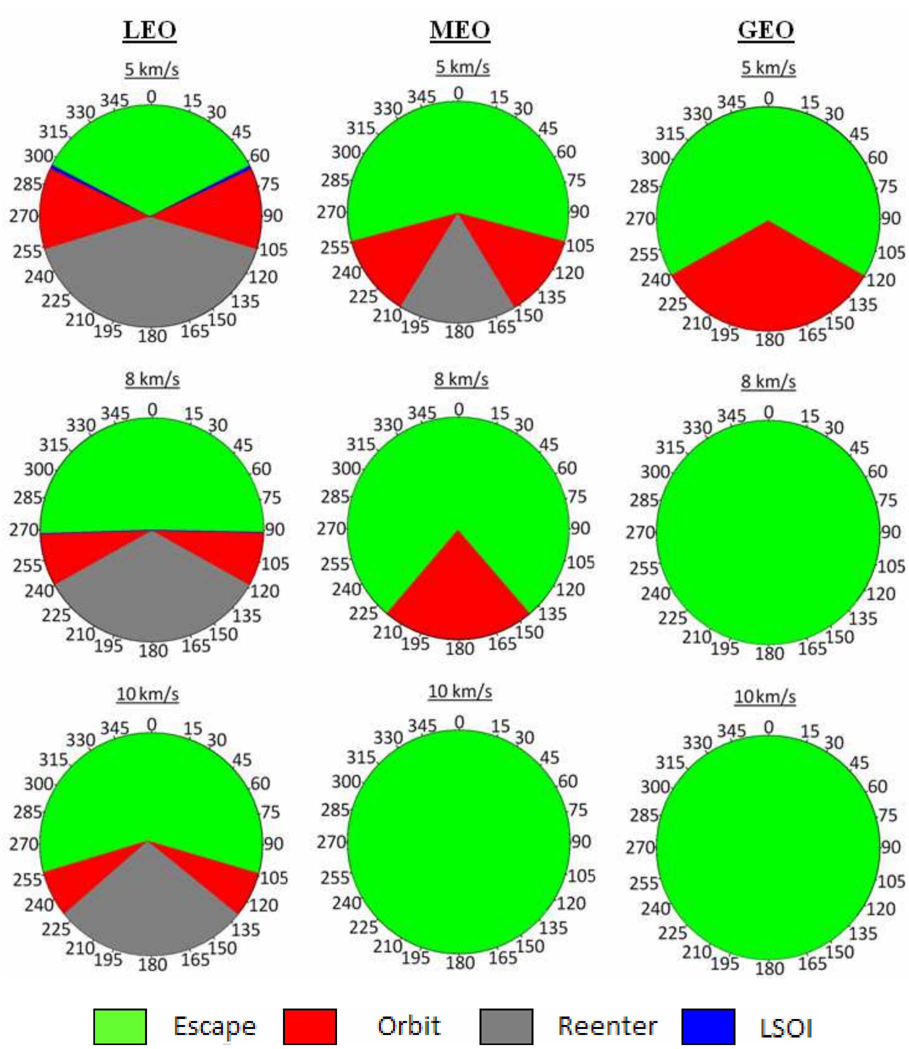

Figure 3. Macron trajectories for all possible exit velocity, firing angle and firing altitude combinations. *0 degrees points in the RAM direction and 180 degrees points in the anti-RAM direction.*

With the exception of a few scenarios, all macrons fired with any component of their velocity in the RAM direction escaped. Furthermore, in LEO, all macrons fired solely in the anti-RAM direction will be place on a reentry trajectory. This will become significant when analyzing possible implementations of the MLP system because it minimizes the level of impact on the orbital debris environment. Firing in the anti-RAM direction is the most efficient direction to maneuver to alter an orbit's size or shape. Likewise, for GEO maneuvers, all macrons fired with an exit velocity ≥8km/s will escape regardless of firing angle, potentially minimizing a macron's level of impact.

C. Orbital Lifetimes

Orbital lifetime calculations were conducted for all firing scenarios which resulted in orbital trajectories. The orbital lifetime of a macron can be viewed as being proportional to its potential impact on the orbital debris environment. Likewise, the greater the orbital lifetime, the more stable the orbit. As with all orbital lifetime predictions, there is an intrinsic amount of error due to inaccuracy in predicting highly variable space weather.[8]

Two orbital lifetime models were created using MATLAB. Both codes were designed to determine the orbital lifetime of a macron. The first code integrated the force due to atmospheric drag throughout an orbit (i.e. integrated drag force code) while the second code utilized the Equations of Variation (i.e. EOV code). In both codes, no third-body or other Earth perturbation forces (e.g. J2) were taken into account. The only external forces applied were the forces due to atmospheric drag and gravity. Macrons were modeled as spheres to directionalize the force of drag in the anti-RAM direction. For simplicity, atmospheric drag was considered to be negligible at orbital altitudes greater than 2,000km.[9] Atmospheric density data was obtained through the Naval Research Laboratory's Mass Spectrometer and Incoherent Scatter Radar model derived from empirical data released in the year 2000 (NRLMSIS-00)[10] for solar minimum, maximum and mean periods. This data was incorporated into both MATLAB drag models on an 11 year solar cycle. The phase of the solar cycle influences the atmospheric density profile which affects the accuracy of an orbital lifetime.[11] Simulations were started with F10.7 values corresponding to solar mean accurate for an Epoch in 2009. Each MATLAB code terminated once the macron's altitude of perigee equaled the radius of the Earth or once the orbital lifetime calculation reached 1,000 years.

The accuracy of the EOV code was compared against the integrated drag force code. Both MATLAB codes produced similar results; however, computational times varied drastically. The computation time required for the integrated force of drag code was several days. Therefore, the EOV code was applied to all subsequent orbital lifetime calculations. The resulting orbital lifetime data from the EOV code is shown in Fig. 4.

An analysis was conducted to determine the optimal time step of the EOV code. Reduced time steps allow for greater accuracy but at the cost of an increased computation time. The parametric time step yielded a standardized time step of 60 seconds based on the accuracy of the results obtained.

D. Macron Size

The force of drag on a macron is a result of the atmospheric density (ρ), macron velocity (v), coefficient of drag (C_d) and cross-sectional area (A). Macron exit velocities were bound between 5 and 10km/s and the overall shape of the macron was defined as spherical; therefore, the only unconstrained variable in the force of drag (F_d) equation was the cross-sectional area. For a given mass, the cross-sectional area can be changed by using a different material or by varying the geometry (e.g. hollow spheres). An increase in a macron's cross-sectional area will decrease the macron's ballistic coefficient, increasing the force due to drag and decreasing its orbital lifetime. The equation for drag is

$$F_d = \frac{1}{2}\rho v^2 c_d A \tag{3}$$

Orbital lifetime data presented in Fig. 4 utilized a constant macron cross-sectional area based on the assumption that each macron was a solid Aluminum sphere with a mass of 1g, yielding a diameter of 0.891cm. The theoretical coefficient of drag for a spherical satellite (C_d = 2) was used throughout all orbital lifetime calculations.[12] In reality, spherical satellites experience a coefficient of drag between 2.2 and 2.4 depending on the orbital altitude.[13] Fig. 5 illustrates the affects of an increased cross-sectional area.

As expected, any increase in the macron's cross-sectional area decreased the resulting orbital lifetime. A decreased orbital lifetime reduces the level of impact on the orbital debris environment. However, an increase in the cross-sectional area of each macron would equate to an overall decrease in propellant storage density.

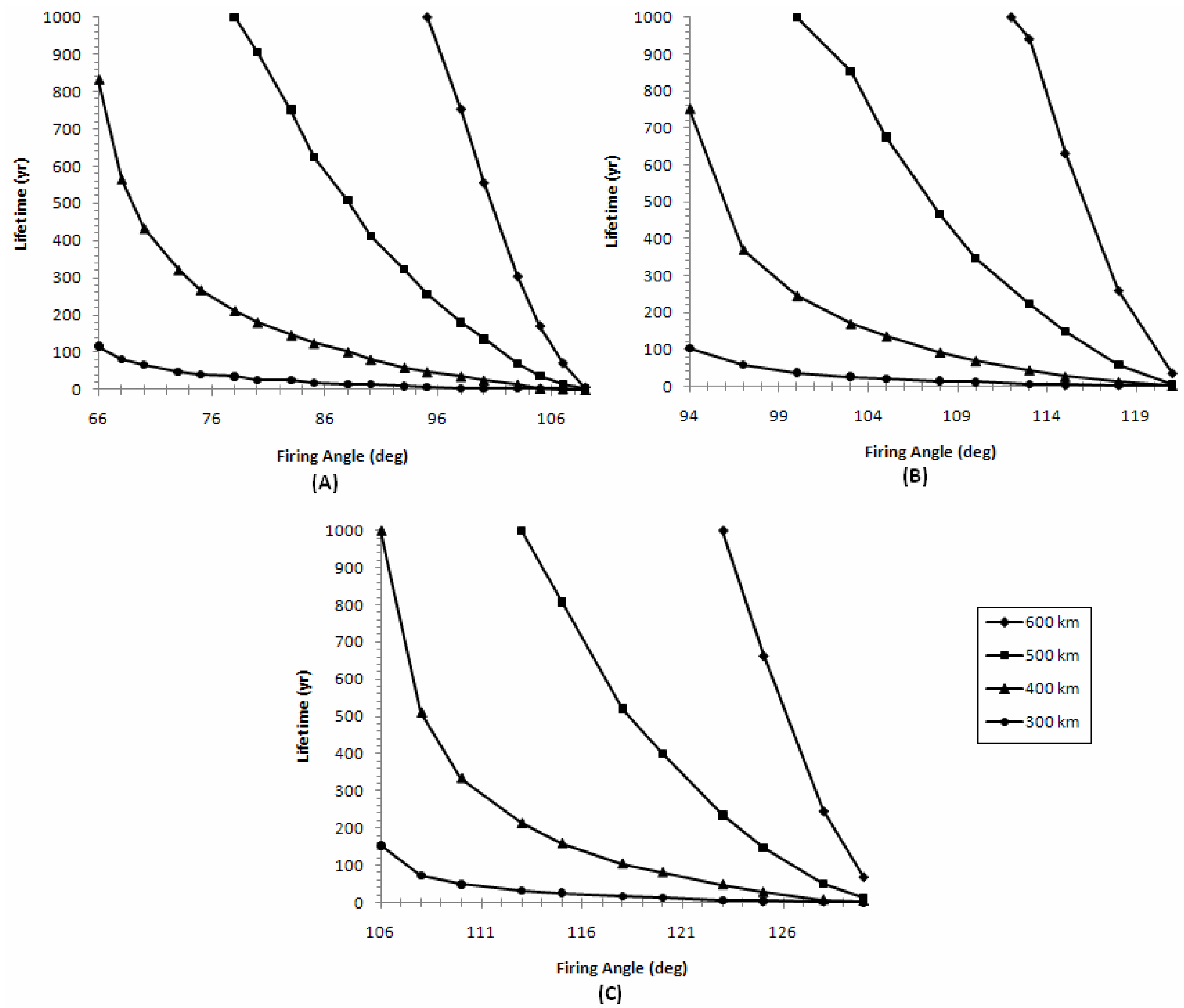

Figure 4. EOV code orbital lifetimes versus macron firing angles as a function of firing altitude.
(A) 1g, 7.07km/s (B) 10g,2.24km/s (C)1g, 10km/s

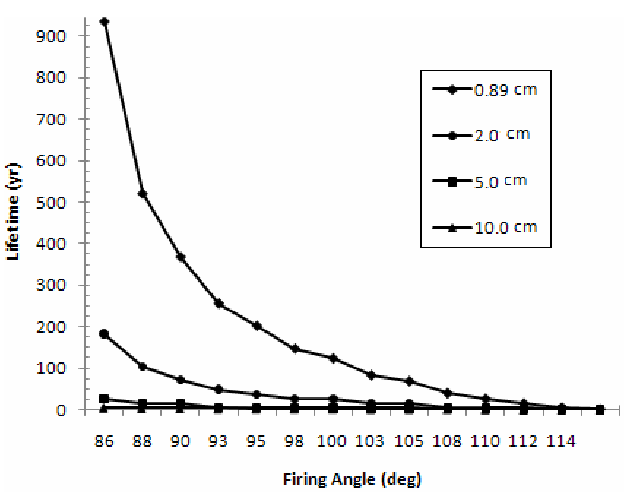

Figure 5. Cross-sectional area affects on orbital lifetimes as a function of macron diameter.

III. Orbital Maneuvers

Three primary orbital maneuvering scenarios were created using STK's Astrogator tool kit to analyze an MLP system's ability to alter an orbit. The three scenarios of interest were intra-LEO transfers (i.e. 400 to 600km altitude), LEO to GEO transfers and GEO inclination changes. For all scenarios, theoretical performance predictions were calculated using Hohmann transfer and Simple Plane Change (SPC) assumptions, respectively. These theoretical performance predictions were compared with the simulated data from STK to provide performance insights.

Three macron mass and exit velocity combinations were held constant throughout all orbital maneuvering simulations as seen in Table 1. An initial spacecraft mass of 1,000kg was also held constant. The mass and exit velocity combination of a macron determines the energy per firing. This particular system was held to a maximum energy per firing of 25kJ; however, one macron mass and exit velocity combination was chosen to simulate a 50kJ firing to quantify the performance increase resulting from an increase in energy. Fig. 6 outlines a general prediction of the number of macrons required to perform a maneuver based on the required change in velocity to complete a maneuver.

KE (kJ)	Mass (g)	Diameter (cm)	Ve (km/s)	F (N)	Isp (s)
25	1	0.891	7.07	7.07	720.8
	10	1.920	2.24	22.36	227.9
50	1	0.891	10.00	10.00	1,019.4

Table 1. Macron exit velocity and mass combinations.

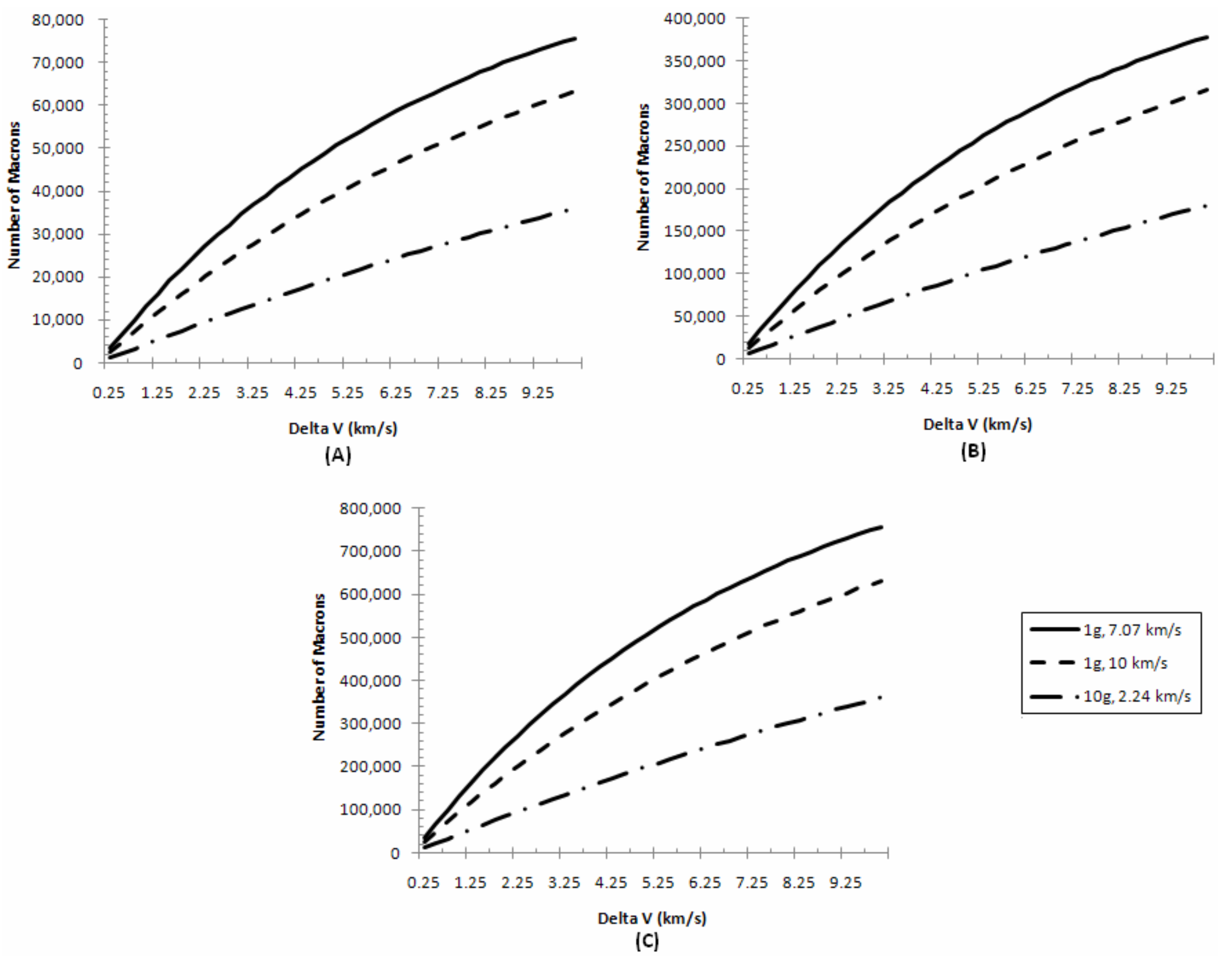

Figure 6. Number of macrons required per delta V as a function of system configuration.
(A) 1g, 7.07km/s, 100kg Satellite (B) 10g, 2.24km/s, 500kg Satellite (C)1g, 10km/s, 1,000kg Satellite

A. Intra-LEO Transfers

Three intra-LEO transfer (i.e. 400 to 600km altitude) simulations were created with STK using the macron mass and exit velocity combinations in Table 1. Each intra-LEO transfer scenario consisted of a series of finite (1,000 to 2,000s) burns centered about either perigee or apogee over a several orbit period. Theoretically, the most efficient way to increase a spacecraft's altitude is to perform a Hohmann transfer. Hohmann transfer calculations utilize coplanar, co-apsidal, instantaneous burn and constant spacecraft mass assumptions. STK scenarios assumed constant spacecraft mass and finite burns. The maneuvering inefficiencies associated with non-impulsive Hohmann transfers were minimized by centering each finite burn over its point of interest (e.g. perigee or apogee) and by minimizing the ratio of burn time-to-overall maneuver time.[14] A comparison can be seen in Table 2 for the number of fired macrons to go from an altitude of 400 to 600km.

Configuration	1g, 7.07km/s	10g, 2.24km/s	1g, 10km/s
Hohmann	15,654	4,950	11,069
STK	14,900	4,900	10,950

Table 2. Numbers of macrons required to complete an intra-LEO transfer (400 to 600km altitude) for a 1,000kg satellite as a function of system configuration.

In theory, a Hohmann transfer calculations represent the theoretical minimum delta velocity (i.e. number of macrons) required to complete a given transfer assuming a constant spacecraft mass. Any error associated with the STK simulations is due to non-perfectly circularized final orbits. All scenarios achieved at least a 99.9% circularized final orbit (i.e. eccentricity $\leq$ 0.001) which accounts for the discrepancies between STK simulations and pure Hohmann transfer predictions. As shown in Fig. 3, macrons fired in the anti-RAM direction are desirable in LEO. This scenario causes a fired macron to have a low relative velocity with respect to Earth after firing which places the macron a direct reentry trajectory, minimizing the impact on the space environment for LEO firings.

In reality, a maneuvering spacecraft will experience mass losses due to propellant usage. This mass loss induces an increased impulse for subsequent firings. A variable satellite mass was not considered for intra-LEO transfers and therefore, the results in Table 2 can be considered a worst-case scenario.

B. LEO to GEO Transfers

The implementation of this system as a LEO (i.e. 300km altitude) to GEO transfer propulsion system was considered. The notional MLP system is capable of operating with varying macron mass, exit velocity and repetition rates. The operational power range of this propulsion system is between 25 to 250kW with an energy per firing of 25kJ. This power level equates to a repetition rate between 1and 10Hz. At these repetition rates, modeling the overall system as either a long duration finite burn or numerous impulsive burns should not have a major effect on the total number of macrons required to complete a transfer. These situations merely represent different system configurations and modeling techniques with identical outcomes. Due to the complexity and long computation times of modeling a LEO to GEO transfer as numerous impulsive burns, the long duration finite burn method was adopted for all subsequent simulations.

Simulations were setup to model a non-constant mass satellite transfer from LEO to GEO. Intuitively, a LEO to GEO transfer which accounts for the mass loss due to propellant usage should require fewer macrons than a constant mass Hohmann transfer. These results are shown in Table 3.

Configuration	1g, 7.07km/s	10g, 2.24km/s	1g, 10km/s
Hohmann	545,203	172,408	385,517
STK	478,400	NA	368,500

Table 3. Number of macrons required for LEO to GEO transfer as a function of system configuration.

The data in Table 3 highlights the reality that a Hohmann transfer calculation result in the maximum number of macrons required to complete a transfer. In addition, the above data highlights a sever performance limitation

encountered by all spacecraft propulsion systems, specific impulse.[15] Specific impulse is a measure of the total impulse per unit weight of propellant and is one of several key performance parameters when analyzing a propulsion system. Of the three macron mass and exit velocity combinations, a 10g macron with an exit velocity of 2.24km/s has the lowest specific impulse at 227.9s, as seen in Table 1. This configuration requires a larger propellant mass than the overall spacecraft mass (1,000kg).

Alternatively, the ideal rocket equation (IRE) can be manipulated to determine the mass that can be transferred from LEO to GEO given a variety of MLP configurations. Fig. 7 shows the number of macrons required to transfer a given overall satellite mass (i.e. all mass excluding propellant mass) from LEO to GEO for both the ideal rocket equation and STK simulations.

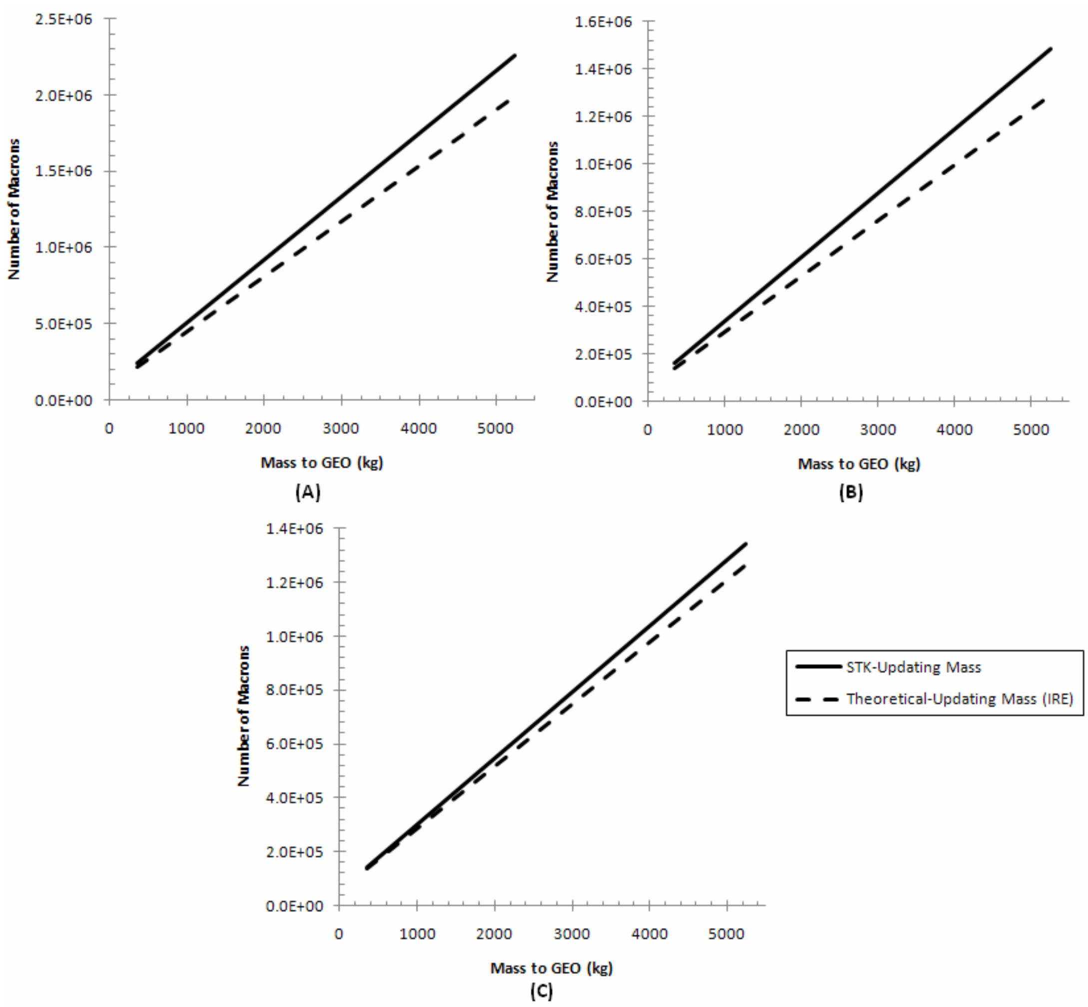

Figure 7. Number of macrons required to transfer a given mass from LEO to GEO.
(A) 1g, 7.07km/s (B) 10g,2.24km/s (C)1g, 10km/s

LEO to GEO transfers require this system to continuously operate throughout the LEO, MEO and GEO regimes. The optimal system configuration to complete a LEO to GEO transfer minimizes the energy per firing, minimized the number of macrons required, and maximizes Isp while avoiding orbital trajectories. According to Fig. 3, the lowest energy configuration which results in the ideal situation of reentry trajectories in LEO and escaping trajectories in GEO requires an exit velocity of 8km/s for a 1g macron. However, in MEO, an exit velocity of 8km/s results in retrograde orbital trajectories. Since the vast majority of satellites are in prograde orbits, retrograde orbital trajectories are extremely hazardous due to the possibility of high relative velocities. Orbital trajectories, while maneuvering through MEO using a 1g-8km/s configuration, can be avoided by either increasing the energy per firing (i.e. increasing the exit velocity) or utilizing off-axis thrusting. Increasing the exit velocity to 10km/s would

result in escaping trajectories for all MEO fired, 1g macrons. Off-axis thrusting requires precise pointing accuracy, slew rate compensation, an increased number of macrons, and knowledge of the resulting trajectory for all firing angles, exit velocities, and altitude combinations. Off-axis thrusting would require a thruster pointing accuracy of less than 1degree and could induce a satellite slew rate. To compensate for satellite slewing, symmetric firings across the RAM axis or a secondary attitude control system would be required. Symmetric firings across the RAM axis requires the MLP system to rotate its firing angle after each firing. Fig. 8 provides trajectory analysis for varying altitudes and firing angles for a 1g macron with an exit velocity of 8km/s.

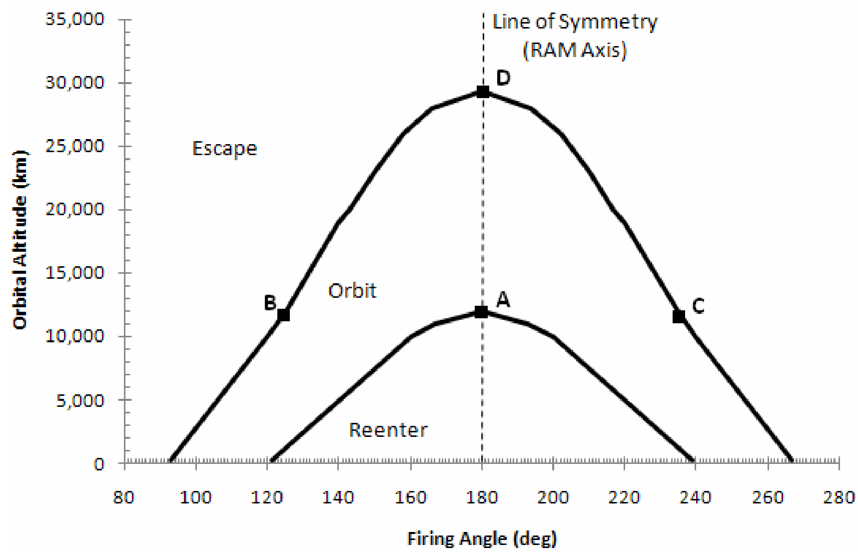

Figure 8. Macron trajectory for LEO to GEO transfer altitudes using a 1g-8km/s system configuration.
Point: A (180deg, 12,004km), B (125deg, 12,004km), C (235deg, 12,004km), D (180deg, 29,382km)

For example, to conduct a LEO to GEO transfer without changing the MLP system configuration (e.g. 1g, 8km/s), the following thrusting sequence is required. Thrust in the anti-RAM direction (i.e. 180 degrees) from LEO up to an altitude of 12,004km (point A in Fig. 8). Off-axis thrusting is required from altitudes of 12,004 to 29,382km (a line connecting points A and D in Fig 8). The most efficient firing angles for off-axis thrusting are the boundary angles represented by the Escape-Orbit line in Fig. 8. These boundary angles maximize the thrust in the anti-RAM direction and minimize off-axis thrust components. The change in firing angle between Points B and C in Fig. 8 is 110deg. In this configuration, the MLP systems must be capable of a firing angle change rate of 110deg/s for a 1Hz system, or two MLP systems can be used to reduce the required firing angle change rates. At an altitude of 29,382km, normal thrusting in the anti-RAM direction may be resumed until GEO is achieved.

Additionally, operating this system under a different system configuration can optimize maneuvering results for a given transfer. Conducting a LEO to GEO transfer using a 10g-2.24km/s configuration would result in all macrons fired in the anti-RAM direction to reenter for all altitudes from LEO to GEO. Increasing a macron's firing altitude for a reentry trajectory can dramatically increase the potential of a macron-satellite collision. A 10g-2.24km/s configuration limits the complexity of a LEO to GEO thrusting sequence at the cost of a decreased Isp as compared to a 1g-8km/s configuration. A decreased Isp results in an increased propellant mass to complete the transfer.

To complete a LEO to GEO transfer thrusting only in the anti-RAM direction, approximately 160,360 macrons would be placed in orbital trajectories. The 2007 Chinese ASAT test resulted in approximately 2 million 0.1 to 1cm diameter and over 40,000 1 to 10cm diameter debris particles.[16]The Iridium-Cosmos collision in 2009 resulted in 1,313 debris particles.[17]

C. GEO Inclination Changes

The last orbital maneuver analyzed was a GEO inclination changing maneuver. Using simple plane change (SPC) assumptions, constant spacecraft mass and instantaneous burns at either the ascending or descending node, a theoretical number of macrons required to complete an inclination change were calculated.

$$\Delta V_s = 2V_i sin\left(\frac{\theta}{2}\right)$$
(4)

These results from Eq. (4) were compared to simulated results created within STK, as seen in Fig. 9. All GEO inclination changing simulations were modeled as long duration, non-constant mass, finite burns centered on either the ascending or descending node over a several orbit period. Small maneuvering inefficiencies increased in magnitude as the burn durations increased, as expected.

STK scenarios were developed using three different repetition rates to verify a convergence of the data to the theoretically computed data. The burn time per ascending or descending node was held constant at 2,000s with varying repetition rates of 1, 5 and 10Hz. A 2,000s burn duration requires the burn to begin 3,072km before either the ascending or descending node and end 3,072km after the node. The overall burn distance is relatively insignificant as compared to the circumference of GEO. As the repetition rates are increased, the overall time to complete a maneuver decreases. Decreasing the repetition rate requires an increase in the number of burns to complete the same transfer equating to an increased number of orbits (i.e. time) to complete the maneuver. The results of varying repetition rates are summarized in Fig. 10.

IV. Conclusion

It is critically important for this system to be utilized in a manner which minimizes its effects on the orbital debris environment. Trajectory analysis shows, in general, macrons which are fired with the majority of their velocity directed in either the RAM or anti-RAM directions, minimize the potential impact on the space environment. Firing angle, exit velocity and altitude combinations which result in orbital trajectories maximize this system's impact on the debris environment and should be avoided if possible. The three most important factors to consider before each firing are the macron's firing angle, exit velocity and the altitude at which the firing will take place. These three parameters combined dictate the level of a macron's impact. Before any orbital maneuvers are conducted, it is critical that spacecraft operators take precautions to ensure avoidance of orbital trajectories or collision course trajectories with neighboring spacecraft.

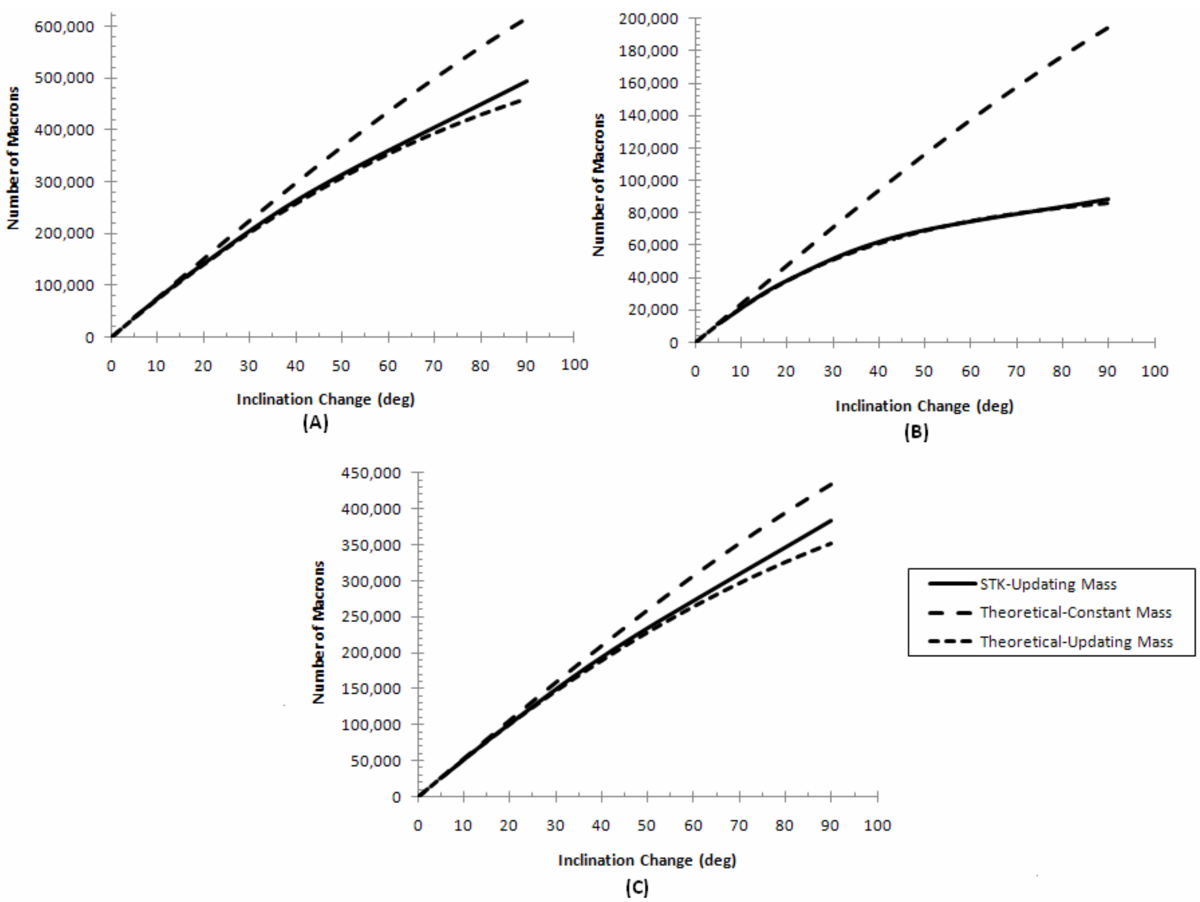

Figure 9. Number of macrons required to complete a given inclination change at GEO.

Orbital analysis supports the implementation of this system as an intra-LEO, LEO to GEO or GEO inclination changing propulsion system. Operating this system at GEO minimizes the potential impact of a macron due to the decreased energy levels required to place a macron on an escaping trajectory. In addition to GEO inclination changing maneuvers, this propulsion system could also be implemented as a GEO phasing or GEO to GEO rendezvous orbital maneuvering system. The data presented in this paper outline the performance parameters of this macron propulsion system and serves as a guide for spacecraft operators to effectively implement and utilize a MLP system.

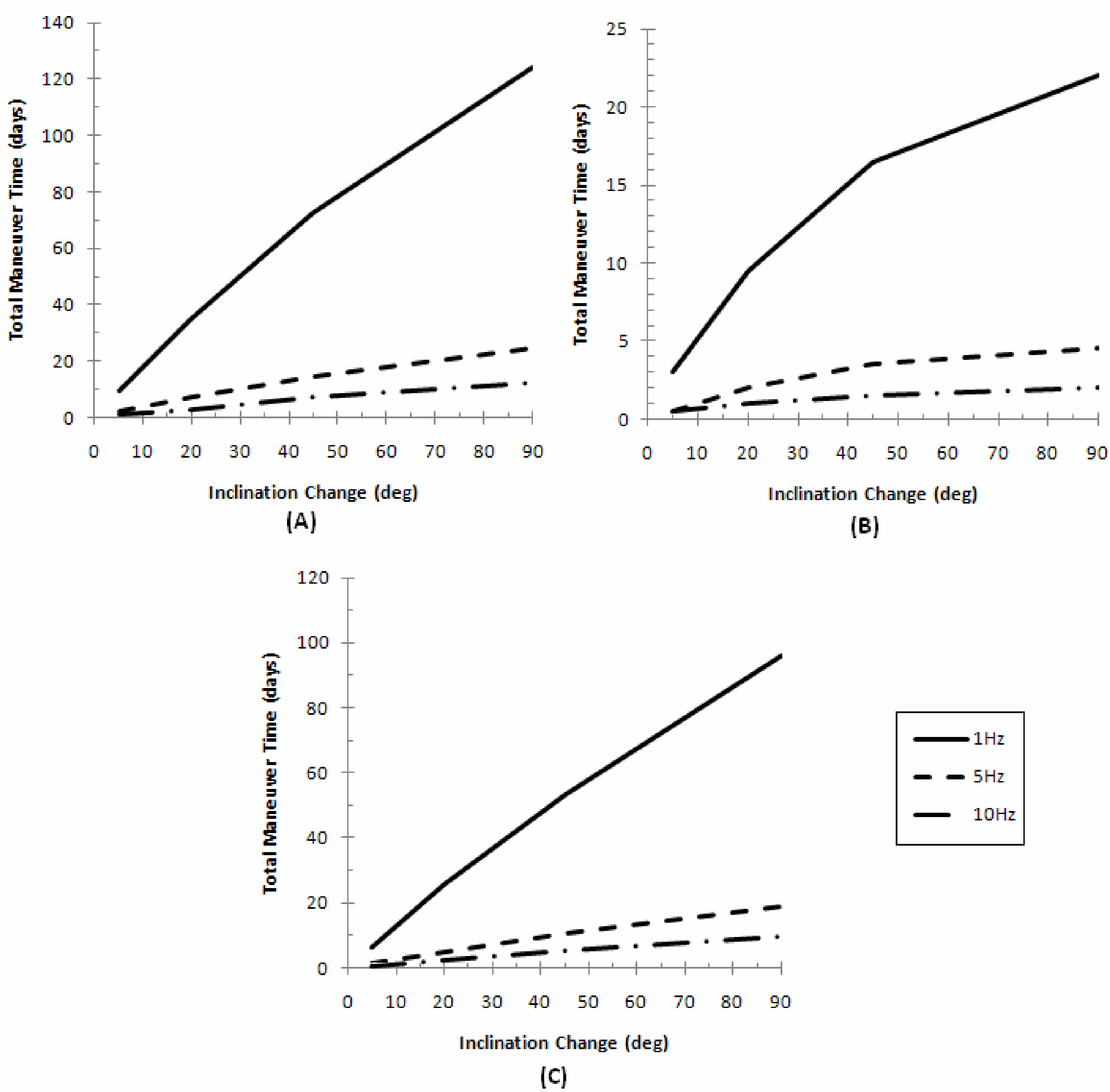

Figure 10. Total time to complete a GEO inclination change as a function of repetition rate.
(A) 1g, 7.07km/s (B) 10g,2.24km/s (C)1g, 10km/s

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